AN OVERVIEW OF THE AEROCAPTURE FLIGHT TEST EXPERIMENT (AFTE)

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ABSTRACT

The Aerocapture Flight Test Experiment (AFTE) is an $80M Earth orbit technology demonstration flight intended to provide an end-to-end validation of low lift-to-drag aeroshell aerocapture technology. A complete Phase A mission and vehicle design was produced during the New Millennium Program's ST-7 competition for which aerocapture was one of the candidate technologies for flight demonstration. Although ultimately not selected for implementation, the AFTE design provides a foundation upon which future aerocapture flight test missions can be considered. The design consists of a 195 kg vehicle comprised of a fully functional spacecraft inside a 70 degree sphere-cone aeroshell. It is launched as a secondary payload to geosynchronous transfer orbit. As the vehicle descends from geostationary altitude, it is targeted into a precise entry corridor in the atmosphere where it performs a guided hypersonic flight. Atmospheric drag will provide a total of 2.4 km/s of velocity decrease leaving the spacecraft in a post-aerocapture orbit with a 400 km apoapsis. A propulsive periapse raise maneuver is subsequently performed to place the spacecraft into a 200 by 400 km parking orbit for several hours. All stored data is transmitted to ground stations during this time, after which the spacecraft is de-orbited into the ocean. The entire flight test lasts for less than one day under nominal conditions. The paper describes the technical details underlying this mission scenario and discusses how the results would benefit future aerocapture missions and related hypersonic flight applications like precision landing.

INTRODUCTION

Aerocapture is one of a class of aeroassist maneuvers that can be used to modulate a spacecraft trajectory with aerodynamic forces when in sufficient proximity to a planetary atmosphere. As the "capture" part of the name denotes, aerocapture uses drag force to decelerate a spacecraft upon arrival at a planet so that its speed drops below escape speed and causes the vehicle to be gravitationally captured into orbit. Thruster firings are required to orient the vehicle during aerocapture, but do not otherwise alter the velocity. This approach contrasts sharply with conventional propulsive orbit insertion where thruster firings provide all of the deceleration and typically consume a significant fraction of the vehicle mass in spent propellant. Aerocapture is also different than aerobraking although both involve drag force deceleration (See Fig. 1 below). Aerobraking is an orbit circularization technique used after an initial high eccentricity orbit has been achieved with propulsive orbit insertion. Multiple small velocity change (ΔV) passes through the atmosphere drop the apoapsis over a period of weeks or months, followed by a small propulsive periapse raise maneuver to complete the orbit circularization. Aerocapture is an orbit insertion technique in which a single atmospheric pass provides enough ΔV to achieve the hyperbolic to elliptical orbit transition. In this sense it mimics the short duration nature of propulsive orbit insertion with the primary advantage of not needing to carry anything more than attitude control system (ACS) propellant.

{Insert Fig. 1 here}

Despite the considerable advantage of propellant mass savings, aerocapture technology has never been attempted in any mission. A partial explanation is that the missions to date have had sufficiently modest orbit insertion requirements so that chemical propulsion technology sufficed. This situation has changed in recent years as the science community moves towards more ambitious planetary missions with more difficult orbit insertion requirements. For example, the NASA Space Science Strategic Plan in 2000[1] identified several high priority missions that required the propellant mass savings afforded by aerocapture technology: Titan Explorer, Neptune Orbiter, Saturn Ring Observer, Mars Sample Return and Venus Surface Sample Return. The importance and timeliness of aerocapture

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has been echoed by the recent IISTP propulsion technology review which placed it in the highest priority category for near-term technology development.[2] The recent Decadal Survey by the National Academy of Sciences likewise identifies aerocapture technology as a high priority for enabling future missions.[3] This overall mission pull has catalyzed efforts to bring aerocapture technology development and flight validation to completion.

There are many elements to aerocapture technology that must be integrated to yield a robust end-to-end vehicle. The most critical requirements stem from the fact that aerocapture involves very high speed flight (5-30 km/s depending on the planet) through an atmosphere which generates substantial aerodynamic forces and intense aerothermal heating. The spacecraft must be protected from these harsh environmental effects with the standard approach being to put the spacecraft inside a protective aeroshell enclosure that combines thermal protection and aerodynamic surface functionality. This approach has been successfully used on atmospheric entry capsules on many past missions including Apollo, Pioneer Venus, Viking, Mars Pathfinder and the Galileo entry probe. Unlike these entry capsules, however, aerocapture aeroshells levy a requirement for guided flight in order to compensate for the expected approach navigation and atmospheric property uncertainties and thereby achieve the desired post-atmospheric orbit.† This requires a lifting aeroshell of a certain minimal lift to drag ratio (L/D) and the ability to orient the lift vector so as to modulate the trajectory. Specialized on board flight software provides the control strategy based on the desired post-aerocapture orbit, properties of the planetary atmosphere and real-time inertial navigation measurements. For each potential aerocapture mission there is an inherent optimization involved between increasing the lift, and hence modulation capability of the vehicle, versus reducing the errors associated with approach navigation and atmospheric property predictions. Other vehicle constraints like aerothermal heating, aerodynamic pressure and deceleration limits also factor into the final design. Proper vehicle design therefore requires careful systems engineering to combine the many disparate elements into a robust solution.

In 2001, NASA's New Millennium Program (NMP) selected aerocapture aeroshell technology as one of the four finalists for its ST-7 flight test opportunity. Its selection was based on the premise that even though aerocapture technology had become mature for small body missions (Mars, Titan, Earth), it would not be adopted for planetary mission use without a successful flight test demonstration. In this sense it paralleled solar electric propulsion technology which was not used on a planetary mission until the successful flight test demonstration on DS-1. The Jet Propulsion Laboratory (JPL) was asked to lead the ST-7 Concept Definition Study for aerocapture and produce a complete design and implementation plan that would be submitted in competition with the other three technologies. The author of this paper served as the lead for that study which was named the Aerocapture Flight Test Experiment (AFTE). In addition to JPL, four other organizations were added to the project by NMP after winning a competitive technology solicitation in each of four separate aerocapture disciplines. The Charles Stark Draper Laboratory was chosen to provide the guidance, navigation and control software required to execute all aspects of the aerocapture maneuver. Lockheed-Martin (LMA) was chosen to design and build the aeroshell according to the aerodynamic and aerothermodynamic design provided by the team of NASA-Langley and NASA-Ames. NASA-Ames was also selected to lead the testing and modeling of the thermal protection system. In addition to its study leadership role, JPL was chosen to provide project management, mission analysis, systems engineering, assembly, integration and test (ATLO) and mission operations.

Unfortunately, NASA did not select AFTE for ST-7 although no technical show stoppers were identified. The need for an aerocapture flight test remains, however, and it is expected that the AFTE design will serve as the starting point for any future attempt to do a low-cost aeroshell aerocapture flight test experiment. The remainder of this paper will describe the details of the AFTE design and its relevance to future planetary missions.

FLIGHT TEST EXPERIMENT OBJECTIVES

The primary objective of the ST-7 Aerocapture Flight Test Experiment (AFTE) is to successfully complete an end-to-end mission that demonstrates all key elements of aerocapture flight and acquires flight data to quantify the vehicle's performance. Table 1 lists the five sub-elements that comprise this primary objective as determined by the AFTE study team. The mission and vehicle designs that meet these objectives are described in the following sections.

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† Apollo was designed for guided hypersonic flight, but this capability was never used.

Table 1. AFTE primary objectives.

	Aerocapture Mission Element	AFTE Quantitative Requirement
1.	Execute a large drag deceleration maneuver with fully automatic guidance, navigation and control to achieve the desired apoapsis altitude.	Minimum 2 km/s ΔV, 400 km apoapsis.
2.	Automatically perform the post-atmospheric propulsive periapse-raise maneuver to achieve the desired final orbit.	Raise periapsis to 150 km.
3.	Integrate the spacecraft into a protective aeroshell enclosure while maintaining full spacecraft functionality.	Continuous 3-axis control, telecom, and thermal management.
4.	Obtain tracking and on board inertial acceleration data to enable post-flight trajectory reconstruction	± 0.5 km (3σ) per axis.
5.	Acquire temperature and pressure data on the surface and interior of the aeroshell to enable characterization of the surrounding flow field and thermal protection system.	Deduce surface convective heat fluxes to within 20%; measure surface pressure to within 10%.

MISSION DESIGN

It became clear early in the study that the AFTE mission objectives could be achieved with a short duration flight in Earth orbit. In particular, the >2 km/s speed decrease requirement could be achieved by using atmospheric drag (aerocapture) to transition from a geosynchronous transfer orbit (GTO) to a low Earth orbit. The flight mechanics of this mission are very close to that of a "true" aerocapture involving a hyperbolic to elliptic orbit change. In addition, this approach requires a substantially reduced propulsion system compared to boosting a vehicle to escape speed and bringing it back to Earth. The potential cost savings are equally large since it is possible to reach GTO as a secondary payload on a communications satellite going to geosynchronous orbit, and thereby avoiding the need for a dedicated launch vehicle.

The complete AFTE mission design proceeded from this basis and is schematically illustrated in Fig. 2. In the baseline mission, AFTE would be launched as an Ariane 5 ECA secondary payload from the Kourou Europe Spaceport in French Guiana. Upon separation from the primary satellite, the AFTE vehicle proceeds out to a GTO apoapsis of 35,780 km in approximately 5 hours. During this cruise period, the spacecraft is in a sun-pointed safe mode with the telecom system transmitting and ground (SN, STSN) and space-based (TDRSS) stations tracking the spacecraft. Navigation updates are provided during this time to correct for launch vehicle injection errors. At the first apoapsis, the spacecraft performs a small deceleration maneuver to lower its periapsis into the atmosphere for aerocapture. There are additional navigation updates to the spacecraft after the apoapsis maneuver to provide a basis for the inertial propagation of the position and velocity during the in-atmosphere part of the flight. The vehicle will enter the atmosphere at 10.3 km/s and exit at 7.9 km/s, giving it a 2.4 km/s decrease. Upon reaching the 400 km apoapsis in the new orbit, the vehicle will execute another propulsive maneuver to raise the periapsis to 200 km, thereby placing the spacecraft into a stable parking orbit. All collected data will be transmitted to ground receiving stations over the next few hours, after which the spacecraft will perform a final propulsive maneuver to de-orbit and drop into the Pacific ocean. The nominal mission duration is 18 hours, although the vehicle has sufficient electrical power to remain in orbit for up to 2 days if problems were to delay either the aerocapture, data relay or de-orbit activities.

{Insert Fig. 2 here}

Figure 3 shows the detailed mission timeline including power consumption, telecom sequence and data flow. Table 2 summarizes the propellant and ΔV budget.

{Insert Fig. 3 here}

Table 2. AFTE ΔV budget

Manuevers	Nominal ΔV (m/s)	Max. ΔV (m/s)	Fuel Load (kg)*
Perigee Lower	19.0	21.3	2.8
Aerocapture	4.0	8.0	1.0
Perigee Raise	51.2	62.6	8.0
De-Orbit	44.7	55.0	6.8
10% Contingency	11.9	14.7	1.8
ACS Allocation	-	-	2.0
Residuals	-	-	0.6
	TOTAL	161.7	22.9

* Based on maximum flight mass of 282 kg (300 kg LV capability – 18 kg of Aeroshell Adapter)

AEROCAPTURE GUIDANCE

The aeroshell provides a fixed lift-to-drag (L/D) ratio which requires bank angle control to change the orientation of the lift vector and thereby modulate the trajectory. The guidance for AFTE is PRED GUID, a Draper Laboratory algorithm originally developed for the Aerocapture Flight Experiment in 1988-1991 but never flown because the program got cancelled. Adapted for use on AFTE, PRED GUID is a numerical predictor-corrector algorithm. Given a current estimated vehicle state from the navigation system, the algorithm performs a numerical trajectory prediction using simplified vehicle and environment models to determine the expected atmospheric exit conditions. The commanded bank angle is then adjusted to null the predicted target apogee altitude miss. Bank reversals are commanded to minimize the wedge angle between the actual orbital plane and the desired target plane. Thus, it is not a reference-following algorithm, which by nature will not use the full capability of the vehicle. With perfect knowledge of the approach trajectory (i.e., no navigation errors), no dispersions in vehicle mass or aerodynamics properties, and nominal atmospheric conditions, PRED GUID will by its nature capture the full entry corridor.

Four degree-of-freedom trajectory simulations were performed for AFTE using Mónte Carlo techniques to factor in uncertainties in approach navigation, atmospheric profiles and vehicle aerodynamics. The post-aerocapture apoapsis delivery results are presented in Figure 4 where the approach trajectory was restricted to an in-vacuum periapse altitude range of 61 to 78 km. The 3σ dispersion is only ±19.8 km centered on an altitude of 398 km which demonstrates the precise and robust nature of PRED GUID for this application.

{Insert Fig. 4 here}

VEHICLE DESIGN

In addition to the aerocapture technology mission objectives listed in Table 1, the AFTE vehicle design was also strongly influenced by the need to minimize cost so as to fit within the constraints of the ST-7 competition. This led to a single-string approach based on high heritage spacecraft components combined with a sufficiently low mass to enable launch as a secondary payload. This approach was feasible because all of the specialized components required for aerocapture could be taken from other spacecraft and atmospheric entry capsule vehicles; therefore, the "new" aspect of AFTE became the systems engineering required to bring these pieces together and use them on an aerocapture application. The last major AFTE vehicle design driver that was also satisfied was maintaining full spacecraft functionality despite enclosing all components inside a blunt body aeroshell. The major affected subsystems are those that require interaction with the outside environment: thermal control, telecommunications, navigation (star trackers/sun sensors) and propulsion. Externally mounted components cannot survive the hypersonic heating environment and therefore must be duplicated internally to be used in the post-aerocapture phase of the mission. This duplication is a mass and cost penalty that detracts from the overall performance on the aerocapture system.

The basic vehicle configuration is shown in Fig. 5. The 1.4 m diameter aeroshell is a Viking/Mars Pathfinder heritage 70° sphere-cone comprised of LMA's SLA 561V/S ablative thermal protection material on top of a monocoque composite honeycomb structure. All internal spacecraft components are mounted on a plate attached to the aeroshell at the heatshield-backshell shoulder joint. The center of mass of the vehicle is offset from the geometric vertical axis to cause the vehicle to fly at a 16° angle of attack, thereby generating a lift-to-drag ratio (L/D) of 0.25. The back plate of the aeroshell includes the launch vehicle adapter ring to give a nose-up launch configuration, an arrangement that aligns the primary launch acceleration with the primary atmospheric drag deceleration. As a cost-saving feature, the aeroshell was designed to not be jettisoned after aerocapture. This functionality is not required for AFTE, although future planetary missions clearly need to remove the aeroshell after orbit insertion.

{Insert Fig. 5 here}

Table 3 is the master equipment list for AFTE showing model numbers, vendors, estimated masses, recommended design contingencies and heritage of components. There is a very large proportion of existing or modified elements, with most of the new elements being structures that are typically customized for a new application anyway. The vehicle is designed to run off battery power only, taking advantage of the short mission duration to avoid the need for externally mounted solar arrays. Other key avionics components include: an RAD 6000 processor on an LMA 6U VME back plane, a Litton LN200 inertial measurement unit, a Ball CT-633 star tracker, Adcole 20020 sun sensors and a Motorola TDRSS 4 S-band transponder. Aeroshell cutouts are provided to enable the star tracker and sun sensor to sense the outside environment, while the transponder communicates through two patch antennas mounted between the TPS and the aeroshell structure, and aligned in opposite directions on the heat shield and backshell respectively. The transponder provides 5 W of transmitted power which results in a data rate of 2-10 kbps to the TDRSS satellites, 50 kbps from geostationary altitude to the ground and 2000 kbps from the post-aerocapture low earth orbit to the ground. The primary instrumentation for AFTE consists of 104 thermocouples embedded in the heat shield and back shell at various locations, interspersed with 24 pressure ports leading to internal sensors (Fig. 6). The data generated by these sensors comprises the bulk of the 300 Mbits total data volume and will be used after the flight to validate the aerothermodynamic models used to design the vehicle.

{Insert Fig. 6 here}

Propulsion for AFTE is provided by 8 - 4.5 N hydrazine thrusters arranged in pairs to provide all attitude control, including up to a 5 deg/s^2 roll rate as required by the aerocapture guidance software. Aeroshell cutouts are provided for each thruster nozzle. The nominal hydrazine load of 19 kg is contained in a titanium propellant tank and supplied to the thrusters with a simple blowdown feed system. The spacecraft thermal control is provided by a combination of elements. White paint is used on the TPS to minimize solar absorption and maximize IR emission. The aeroshell structure itself at the back plate location will serve as an effective radiator for internal heat dissipation despite the presence of TPS material on the external surface. A loop heat pipe will be used to transport heat to this radiator. During the atmospheric portion of the flight, however, the heat pipe will be turned off to limit reverse heat flow back into the spacecraft. The heat soak-back through the aeroshell after aerocapture is passively accommodated by the thermal inertia of the vehicle which limits the temperature rise of critical components to acceptable levels. This approach avoids the cost and mechanical complexity of an aeroshell separation mechanism which would otherwise be required to limit the soak-back problem.

Table 3. AFTE Master Equipment List (MEL)

Subsystem Component	Vendor Product	Heritage	Use	Units Flt	Sp	EDU	Total CBE	Cont	Total Mass
Star Tracker	Ball CT-633	NEAR, ACE, Galex	As-is	1	0	1	2.50	2%	2.55
Sun Sensor Head	Adcole 20020	MAP. MSX	As-is	1	0	0	0.26	2%	0.27
Sun Sensor Elec.	Adcole 20020	MAP, MSX	As-is	1	0	0	1.13	2%	1.15
Inertial Meas. Unit	Litton LN200	DS1, Clementine	As-is	1	0	0	0.75	2%	0.77
DC-DC Converter	Lambda ART2800	AXAF, SIRTIF	As-is	1	0	0	0.18	2%	0.18
Flight Processor Card	LM Fed Sys	MSP '98, Stardust	As-is	1	0	1	1.18	2%	1.20
Mass Memory Card	SEAKR/LMA – 2Gb	MSP '98, Stardust	As-is	1	0	1	1.02	2%	1.04
UL/DL Card	LMA	MSP '98, Stardust	As-is	1	0	1	0.47	2%	0.48
Master I/O Card	LMA	MSP '98, Stardust	As-is	1	0	1	0.45	2%	0.46
PACI Card	LMA	MSP '98, Stardust	As-is	1	0	1	0.50	2%	0.51
Instrument I/O Card	LMA	None	New	1	0	1	0.50	30%	0.65
DC-DC Converter	LMA	MSP '01	As-is	1	0	1	1.00	2%	1.02
VME Backplane	LMA	MSP '98, Stardust	Mod	1	0	1	0.90	20%	1.08
6U Chassis	LMA	MSP '98, Stardust	Mod	1	0	0	1.97	20%	2.36
Power Distribution Box	Litton – PCDU	Image, EO-1, MAP	Mod	1	0	0	2.72	15%	3.13
Battery (315 A-hr)	Saft – LSH20	Deep Impact*	Mod	1	1	2	37.60	10%	41.19
S-Band Transponder	Motorola- TDRSS 4	Balloons, EOS-AQUA*	As-is	1	0	1	4.08	2%	4.16
S-Band Components	Various	Various	Mod	1	0	0	2.30	15%	2.76
Ti Fuel Tank	PSI - 16.5" Sphere	Pathfinder	As-is	1	1	0	5.77	2%	5.89
Thrusters, 4.5N	GD/Primex - MR111	Various	As-is	8	1	0	2.64	2%	2.69
LP Transducer	GP:50	Cassini	As-is	2	1	0	0.54	2%	0.55
Temperature Sensor	JPL – PRT	Cassini	As-is	10	2	0	0.30	2%	0.31
Prop Components	Vacco/JPL	Various	Mod	1	0	0	2.94	23%	3.61
MLI/Coatings	JPL	Various	As-is	1	0	0	1.86	30%	2.42
One-Way Heat Pipe	AeroThermal/Swales	Hughes HS702	As-is	1	0	0	2.25	2%	2.30
Radiator	JPL	None	New	1	0	0	7.01	30%	9.11
Thermal Components	Various	Various	Mod	1	0	0	1.70	30%	2.20
Bulkhead Plate	JPL	None	New	1	0	0	4.60	30%	5.98
Fasteners/Brackets	JPL	Various	New	1	0	0	5.03	30%	6.54
Ballast	JPL	None	New	1	0	0	5.50	30%	7.15
Heatshield Structure	LMA	Pathfinder	Mod	1	0	0	7.82	20%	9.39
Heatshield TPS	LMA – SLA561V	Viking	As-is	1	0	0	14.85	10%	16.34
Backshield Structure	LMA	Pathfinder	Mod	1	0	0	8.32	20%	9.99
Backshield TPS	LMA – SLA561S	Pathfinder	As-is	1	0	0	9.91	10%	10.90
Close-out Plate	LMA	Pathfinder	Mod	1	0	0	2.31	20%	2.78
Misc Hardware	LMA	Various	Mod	1	0	0	4.69	20%	5.63
Thermocouples	Pico/Omega – Type K	Various	As-is	104	96	0	9.43	2%	9.62
Foreshell LP Transducer	Ames	Viking, Shuttle	Mod	11	1	0	0.61	15%	0.70
Transducer I/F	GP:50	None	Mod	2	1	0	1.00	15%	1.15
Backshell LP Transducer	Ames	Viking, Shuttle	Mod	13	1	0	3.90	15%	4.49
Thermocouple Packaging	LMA	Pathfinder, Galileo Probe	Mod	31	1	0	0.31	30%	0.40
Total Spacecraft Dry Mass							175.7		201.94
Hydrazine							19.0		22.9
Aeroshell Adapter							15.2	20%	18.3
TOTAL LAUNCH MASS							**209.91**	**16%**	**243.1**
Launch Vehicle Capability									**300**
LAUNCH MASS MARGIN									**30%**

* Flight Units have been developed and/or are undergoing testing for these developments, even though these projects have not flown.

COST AND MANAGEMENT

The proposed organizational structure for AFTE was based on JPL's flight project organization template, and is shown in Figure 7. Key leadership positions below the system-level managers are identified, including Project Element Managers (PEMs), and Cognizant Engineers (CEs), the PEM designation being granted for larger project elements requiring substantial management responsibility. Under the premise of a NMP ST-7 mission, the JPL-based project office would have executed and managed contracts and other appropriate funding mechanisms (e.g., bypass funding to NASA centers) with all team members. This included contracts to Charles Stark Draper Laboratory (CSDL) for Guidance, Navigation, and Control (GN&C) development, and Lockheed Martin Astronautics Operations (LMA) for aeroshell development. JPL would have provided all financial and other project status reporting to NASA. Management of the AFTE project would have been conducted in accordance with the policies and requirements specified by NPG 7120.5a/b, and JPL's institutional procedures for 7120.5a/b implementation.

{Insert Fig. 7 here}

A grass-roots cost estimate was generated for AFTE based on a Work Breakdown Structure comprised of all spacecraft elements. The total estimated cost excluding the launch vehicle is \$79.4M as summarized in Table 4. This corresponded to a nominal 39 month schedule with Project Formulation Refinement (Phase B) starting in February 2002, launch in January 2005 and completion of post-flight data analysis in April 2005. It should be noted that although AFTE is only a 1 day mission, the ability to exploit this feature for cost savings is very limited given the design approach of using robust, high-heritage spacecraft avionics components. Similarly, the spacecraft software shares most of the functionality characteristic of longer missions which also limits cost saving opportunities.

Table 4: Total AFTE mission cost by WBS element.

	Total (Real Yr. \$K)
Aerocapture Flight Test Experiment (AFTE)	
1.0 Project Management	\$2,454
2.0 Aerocapture Technology	\$1,231
3.0 Mission & Project Systems Engineering	\$3,494
4.0 Mission Assurance & Safety	\$2,255
5.0 Flight System	\$44,459
5.1 Flight System Engineering	\$709
5.2 Spacecraft Avionics	\$26,520
5.3 Spacecraft Mechanical	\$17,230
6.0 Assembly, Test and Launch Operations	\$7,157
7.0 Ground Data System & Mission Operations	\$1,954
8.0 Launch Vehicle	\$0
9.0 Education and Public Outreach	\$586
Total Project Reserves	\$15,791
Total	\$79,382

TECHNOLOGY INFUSION TO FUTURE MISSIONS

The AFTE flight parameters are representative of the class of small body aerocapture missions in the solar system as summarized in Table 5. The commonality of deceleration ΔV, heating rates and 3σ entry corridor widths suggests that the low-L/D blunt body aeroshell approach will be suitable for all of these missions. In contrast, the parameters of gas giant planet missions, as represented by Neptune in Table 5, are significantly different in terms of heating and, on the basis of preliminary analysis, entry corridor widths. This suggests that higher L/D vehicles with different thermal protection systems will be required for these missions. However, the bank angle control guidance strategy and the trajectory and aerodynamic simulation tools are expected to be directly applicable to small bodies and gas giants alike. The other critical aspect of AFTE infusion to future missions is the design methodologies and tools, systems engineering and trained personnel that will be developed and validated with a successful flight test experiment.

Table 5: Comparison of AFTE Flight Parameters with Future Low-L/D Aerocapture Aeroshell Missions

Planet	Entry Speed	Atmospheric Composition	Decel. ΔV	Estimated Peak Heating	3σ Entry Corridor	Achievable Targeting Accuracy
	(km/s)		(km/s)	(W/cm2)	(deg)	(deg)
Earth (AFTE)	10.3	79% N2, 20% O2, 1% Ar	2.4	200	0.8	0.2
Venus	11.5	97% CO2, 3% N2	4.0	400	0.8	TBD
Mars	5.5	95% CO2, 3% N2, 2% Ar	2.0	200	1.0	0.4
Titan	6.5	98% N2, 2% CH4	5.0	500	3.0	1.8
Neptune	30.0	80% H2, 19% He, 1% CH4	6.0	2000	~0	TBD

Although AFTE is focused on aerocapture mission applications, there is a very high degree of overlap with atmospheric entry missions that feature guided hypersonic flight for precision landing purposes. Specifically, both applications could use automatic bank angle control of an aeroshell vehicle with similar aerodynamics because of the shared entry speeds, L/D ratios and hypersonic Reynolds and Mach numbers. The vehicle design methodologies and systems engineering will be likewise be common. Therefore, it is expected that AFTE flight test results will directly support future precision entry and landing missions in these technical areas. It is worth noting that there exists an opportunity to modify the final stage of the AFTE mission to include guided flight from de-orbit to splashdown and therefore directly mimic all flight phases of a precision landing mission. This option was not evaluated in detail during the study, but it seems likely that the impact on vehicle design and cost would be minimal.

CONCLUSIONS

The Aerocapture Flight Test Experiment (AFTE) was developed but not selected by the New Millennium Program for its ST-7 flight test opportunity. Nevertheless, the AFTE design provides a foundation upon which future aerocapture flight test missions can be considered. It consists of a 195 kg vehicle comprised of a fully functional spacecraft inside a 70 degree sphere-cone aeroshell. It is launched as a secondary payload to geosynchronous transfer orbit, from which it executes a 2.4 km/s drag deceleration flight in the atmosphere upon first periapse. Although not strictly an aerocapture mission in the sense of performing a hyperbolic-to-elliptical orbit change, the flight mechanics and 2.4 km/s ΔV of AFTE are relevant. The single-string vehicle design is based on high heritage components and careful systems engineering that produce a robust aerocapture technology flight experiment. The key technical details and mission parameters have been described herein.

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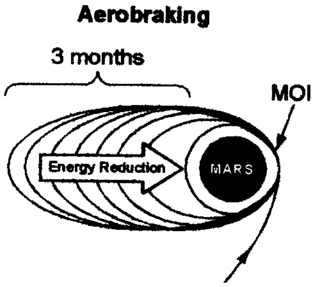

	Aerobraking	Aerocapture
Time to final orbit	**3 Months**	**3 Hours**
MSR S/C (kg)	900	900
Propellant (kg)	700	20
Aeroshell (kg)	N/A	180
Total arrival mass (kg)	**1600**	**1100**

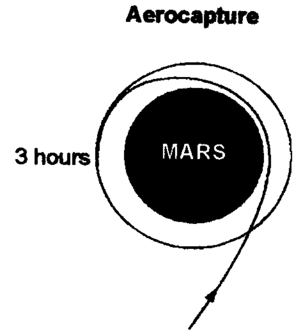

Figure 1. Comparison between aerocapture and aerobraking for candidate Mars orbiter.

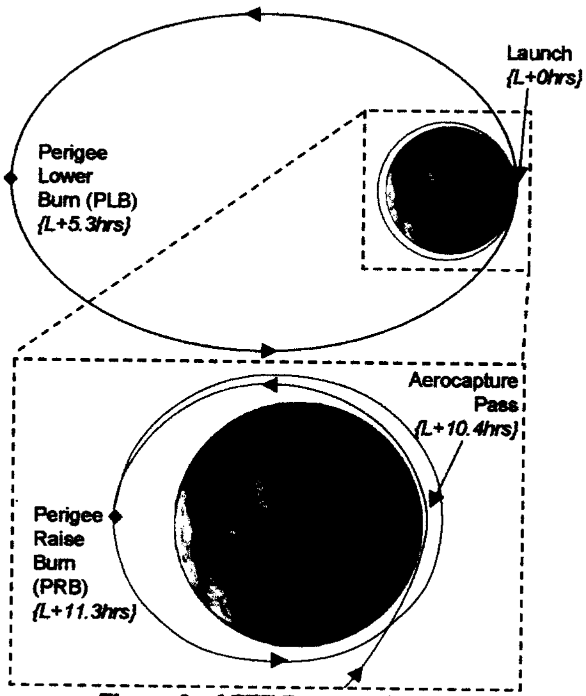

Figure 2. AFTE Baseline Mission.

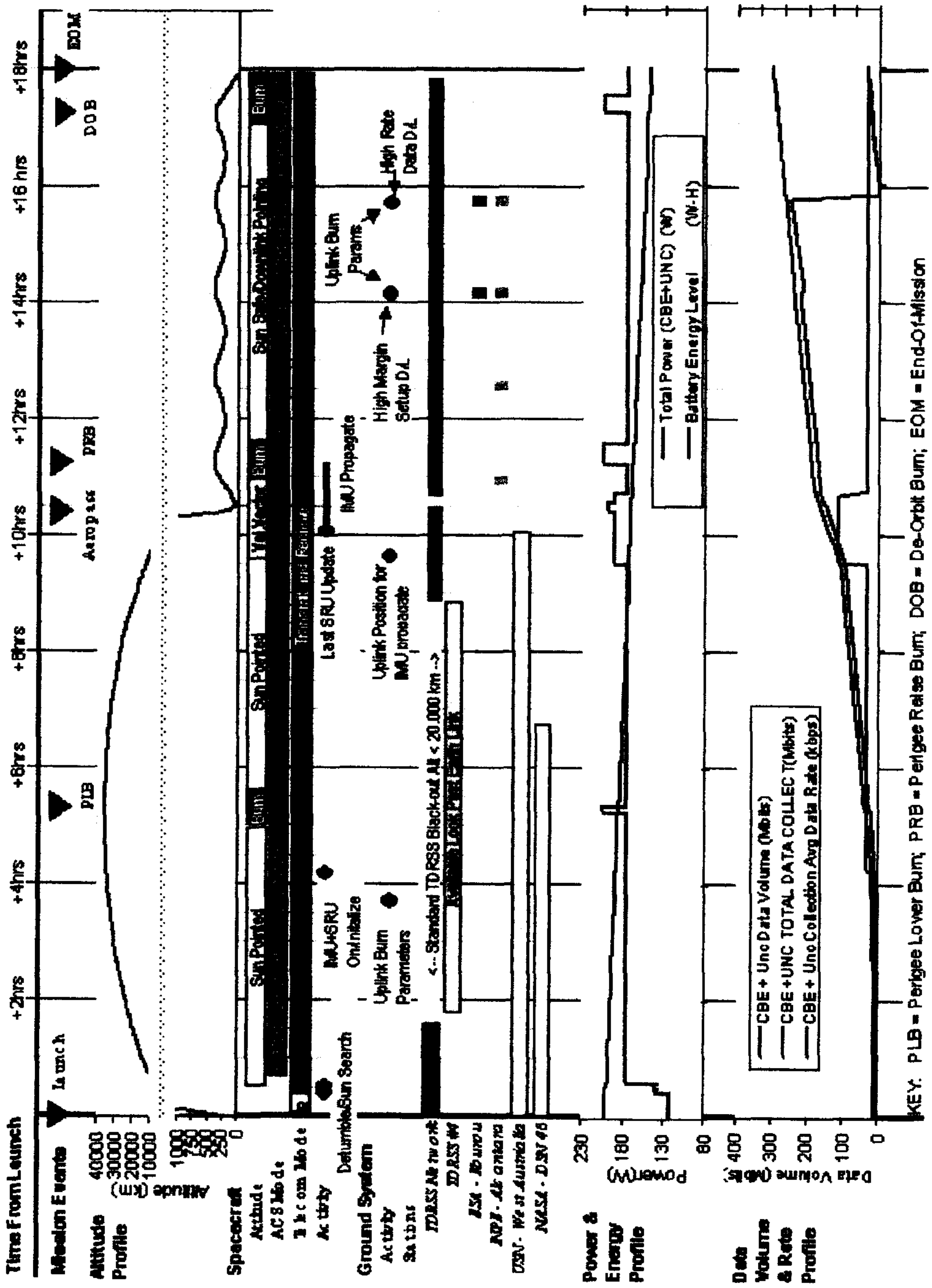

Figure 3. Overall Mission Timeline detailing Stations in view and Power and Data Profiles

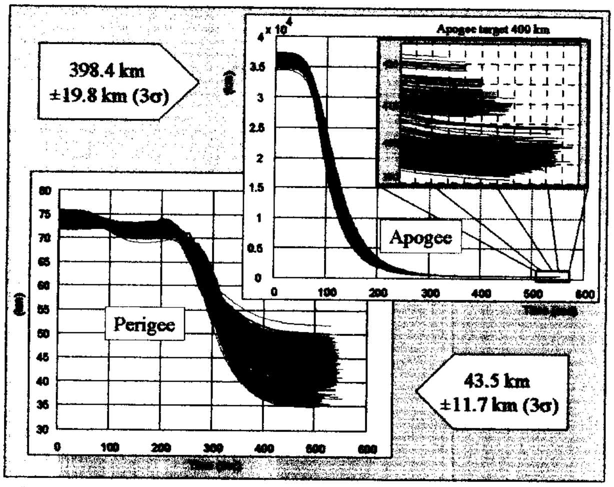

Figure 4. AFTE 4DOF Monte Carlo Results showing altitude versus time trajectories.

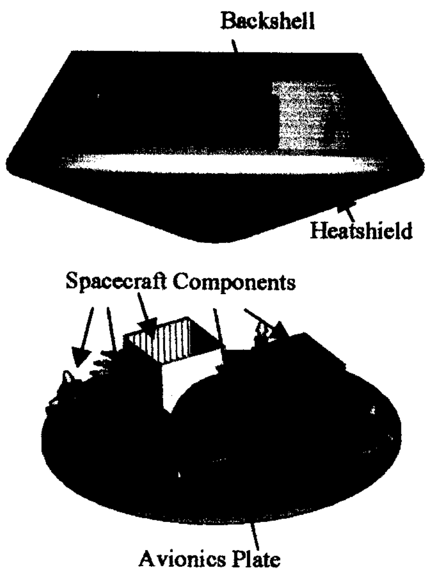

Figure 5. Spacecraft configuration.

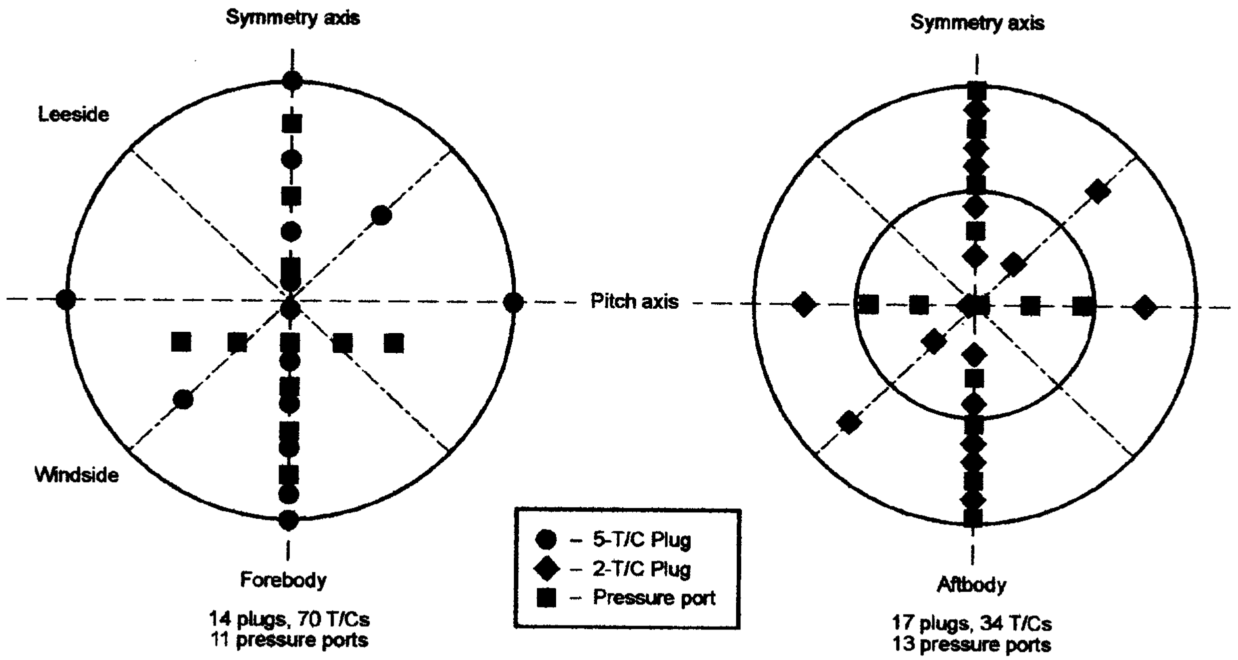

Figure 6. AFTE instrumentation layout.

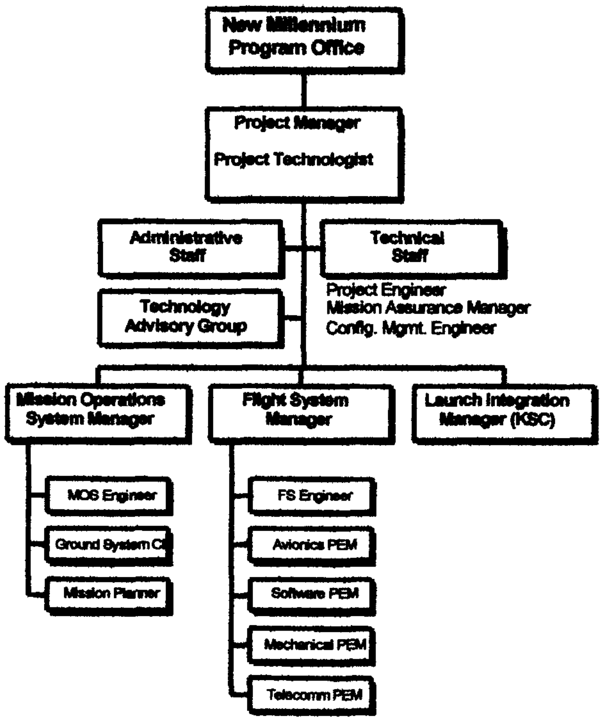

Figure 7. AFTE organizational structure.